Title: *Recent Concept Study for Cryogenic Fluid Management to Support Opposition Class Crewed Missions to Mars*

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Abstract

NASA recently completed a mission concept study to evaluate the feasibility and propulsion technology development requirements for reduced travel duration crewed missions to Mars. A high-level goal of the study was to minimize the health impact on the crew caused by the space environment. This was implemented in the study by limiting the crew to a total of approximately two years of in-space operations and travel time. For the initial mission, the crew would stay about 30 days on the Martian surface. The propulsive demands of such a mission are immense, and the study identified two advanced propulsion options with the potential to meet the mission requirements—both options rely on nuclear fission to provide efficient propulsive energy. One propulsion option was a nuclear electric propulsion (NEP)/Chem Hybrid, with a reactor and energy conversion system powering xenon propellant ion thrusters to provide an efficient, but lower-thrust, push for most of the mission duration. This concept also relied on a liquid oxygen/liquid methane (LO_2/LCH_4) chemical propulsion stage to provide high thrust for maneuvers while near the Earth and Mars. The second propulsion option was nuclear thermal propulsion (NTP), in which the reactor heats liquid hydrogen (LH_2) propellant to expand through a nozzle for thrust at about twice the efficiency of the best chemical propulsion systems. Both vehicle concepts rely on storing large amounts of cryogenic propellant (either LO_2/LCH_4 or LH_2) for multiple years in space without loss, far exceeding state-of-the-art capability. To enable this new capability, the team assumed the use of several advanced cryogenic fluid management (CFM) technologies and analyzed the integrated system performance. This included considering the vehicle-level effects of the size, mass, and power requirements of these CFM elements. Further, the team evaluated the development required to enable such a mission in the mid-2030s and determined that it was feasible. The paper elaborates on the assumed CFM technologies, provides key analysis results, and illustrates the feasibility of technology development for the proposed solutions to the CFM challenges for each propulsion concept.

Keywords: Cryogenic propellant, Zero boil-off, Mars mission vehicles, Nuclear space propulsion

1. Introduction

Two concurrent engineering teams at NASA have recently conducted conceptual design studies to assess the feasibility of quicker “opposition-class” crewed missions to Mars. Although these opposition-class missions are challenging from a propulsive energy perspective, the potential to reduce mission and crew risk by shortening mission duration is highly desirable, particularly for early crewed Mars visits. These missions are limited to approximately two years’ duration and would include 30-day surface stays for the crew. Lander and ascent stages to transport the crew from Mars orbit to the surface and back were studied separately and are not discussed in this paper but were treated as cargo to be delivered to Mars well in advance of crew arrival.

These short trip-time Mars missions are demanding of Earth-to-Mars propulsion; they require on the order of twice the change in velocity (delta-V) compared with a three-year “conjunction-class” mission. Opportunities for opposition-class missions occur roughly every two years, and the delta-V varies with each opportunity. For this study, the 2039 opportunity was chosen as the design point, as the delta-V would bound most of the opportunities in a 15-year period beginning in 2035. The transport vehicle for these missions must deliver significant mass to Mars orbit, including the crew, supporting cargo, and more than two years’ worth of crew consumables. The combination of a high-energy mission trajectory and a large amount of cargo can result in a large and costly transport vehicle. Two advanced, high-performance propulsion technologies were evaluated. The first was a hybrid of nuclear electric propulsion (NEP) and chemical propulsion, evaluated by the Compass team at NASA Glenn Research Center; the second was nuclear thermal propulsion (NTP), evaluated by the Advanced Concepts Office (ACO) at NASA Marshall Space Flight Center. Additional description of the overall mission assumptions can be found in reference 1.

Each approach used cryogenic propellants and required cryogenic fluid management capabilities far beyond today’s state of the art. Several launch vehicle upper stages operate reliably in space now with cryogenic propellants, but the Mars missions studied have three key differences: mission duration, microgravity propellant management, and in-space propellant transfer. Launch vehicle upper stages operate for only a few hours and can tolerate significant propellant loss rates due to boil-off or leakage; upper stages leverage small thrusters to create acceleration that settles the propellant (i.e., separating liquid and vapor) to enable safe vapor venting and liquid supply to engines; and lastly, tank-to-tank cryogenic propellant transfer (i.e., refueling) has never been demonstrated in space. In contrast, for the Mars transportation vehicles studied, the propellant must be stored with minimal loss for multiple years, requiring low to zero heat load and ultra-low leakage; thruster operations to manage liquid location are inefficient over a multi-year mission; and propellant transfer between tanks is required for fueling or vehicle operations. Thus, advanced approaches to the cryogenic propellant system design were needed.

To compare the two propulsion options, it was necessary to provide the concurrent engineering teams with consistent sets of initial assumptions. For example, mission level assumptions included the payload mass (principally a Deep Space Habitat and supplies for the crewed mission), that the crew vehicle will depart from Earth orbit, the targeted mission duration, the number of the Space Launch System launches that could be used, and the range of mission

opportunities to be evaluated [1]. These missions are not anticipated to start before the mid-2030s, so it was reasonable to incorporate lower-maturity technologies into the initial assumptions and take advantage of improved projected performance capability. Both the ACO and Compass teams worked from these initial assumptions, evolving the design if necessary to meet mission performance. In both concept vehicles, the cryogenic fluid management (CFM) systems design was successfully closed with little modification of initial assumptions. This article summarizes the design approach for the respective CFM systems that satisfied the mission requirements for each conceptual vehicle. Further, a proposed approach to complete maturation of the needed advanced CFM technologies in time to support the Mars vehicle design process is presented, demonstrating feasibility of the respective CFM systems.

2. Concept Vehicles

2.1. NEP/Chem Hybrid Vehicle Concept

Figure 1. shows the NEP/Chem hybrid vehicle [1]. The vehicle contains the NEP module, the Xenon Interstage, the crew habitat, and a large liquid oxygen/liquid methane (LO_2/LCH_4) chemical stage. The addition of the chemical stage to the NEP mission served multiple purposes. It significantly reduced the size of the NEP system that is required to meet the two-year mission duration target. This reduction of the NEP system size, in turn, enabled the entire radiator to be packaged in a single launch avoiding in-space assembly of that complex system. Finally, using the high thrust chemical propulsion system to depart from and to capture into both Earth and Mars orbits reduced the “gravity well” losses associated with those maneuvers [1]. The various elements would be separately launched and assembled in orbit. The NEP module combines a nuclear reactor, power conversion, and electric propulsion systems. For power, a 1.9 MWe low enriched uranium (LEU) reactor is used with a lithium hydride/tungsten radiation shield. The reactor is integrated with four 500 kW Brayton convertors along with 2500 m^2 of deployable radiators to dispose of waste heat from the power conversion. Solar arrays and batteries are included for commissioning power. For propulsion, clusters of Hall-effect ion thrusters are used, with xenon as the NEP propellant. Meanwhile, the reaction control system (RCS) is based on heritage Earth storable propellant thrusters. A deployable boom provides separation between the reactor and the rest of the NEP module. Radiators and thruster booms are also deployable.

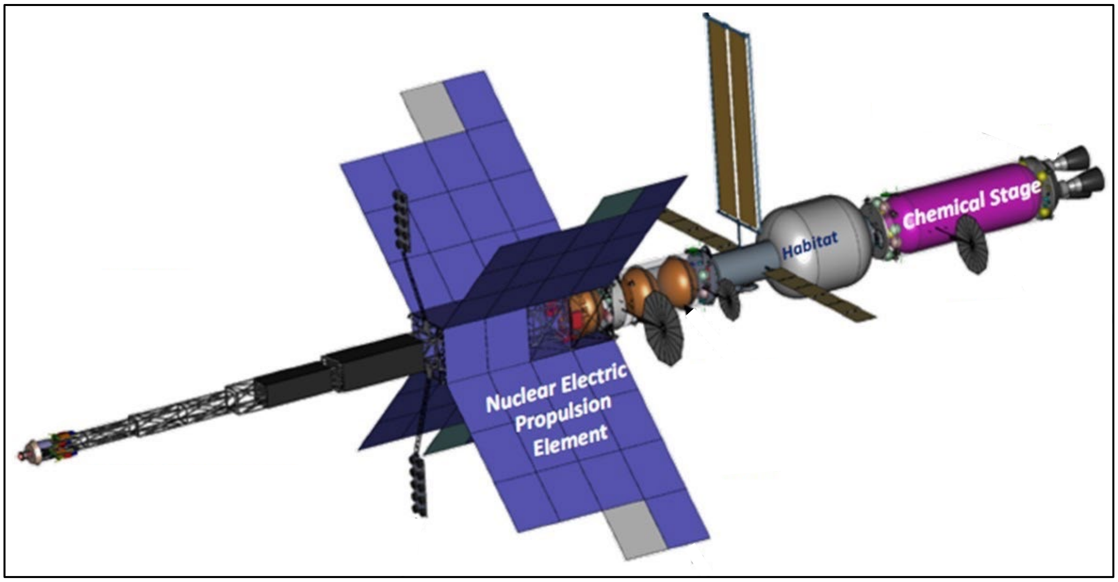

Figure 1. NEP/Chem Hybrid Propulsion Mars Transportation Vehicle Concept

The Xenon Interstage is launched separately and stores xenon propellant in composite overwrap pressure vessels near 300 K. Additional solar arrays and batteries power the Interstage until after docking and reactor startup. The chemical stage is also launched separately, but partially fueled to maximize tank volume within the launch vehicle and will be described in section 4. This stage carries approximately 200 tons of LO_2/LCH_4. Solar arrays and batteries also power the chemical stage until after docking and reactor startup. Storable propellants are also used for the RCS system.

2.2 NTP Vehicle Concept

The NTP vehicle concept, as configured for Earth departure, is shown in Figure 2. The vehicle consists of the core stage, an in-line tank, several drop tanks, and the crew habitat. The various elements would be separately launched and assembled in orbit. The core stage includes the nuclear thermal rocket engines with LEU reactors, a radiation shield, and a large liquid hydrogen (LH_2) tank. The inline tank is a larger, SLS launched LH_2 tank, and the drop tanks would be commercially launched and integrated on a truss structure. As their name indicates, drop tanks are disposed of as the LH_2 is consumed to reduce vehicle dry mass for subsequent maneuvers. Both the in-line and drop tanks have their own reaction control propulsion to facilitate aggregation and assembly. Solar arrays are included for system electrical power. Two additional NTP stages (combining a side-core stage and a side-inline tank) would be mounted to the sides of the core stage to transport the vehicle from its elliptical assembly orbit to the lunar-distant high-Earth orbit from where it will depart to Mars. These stages are disposed before the trans-Mars injection burn.

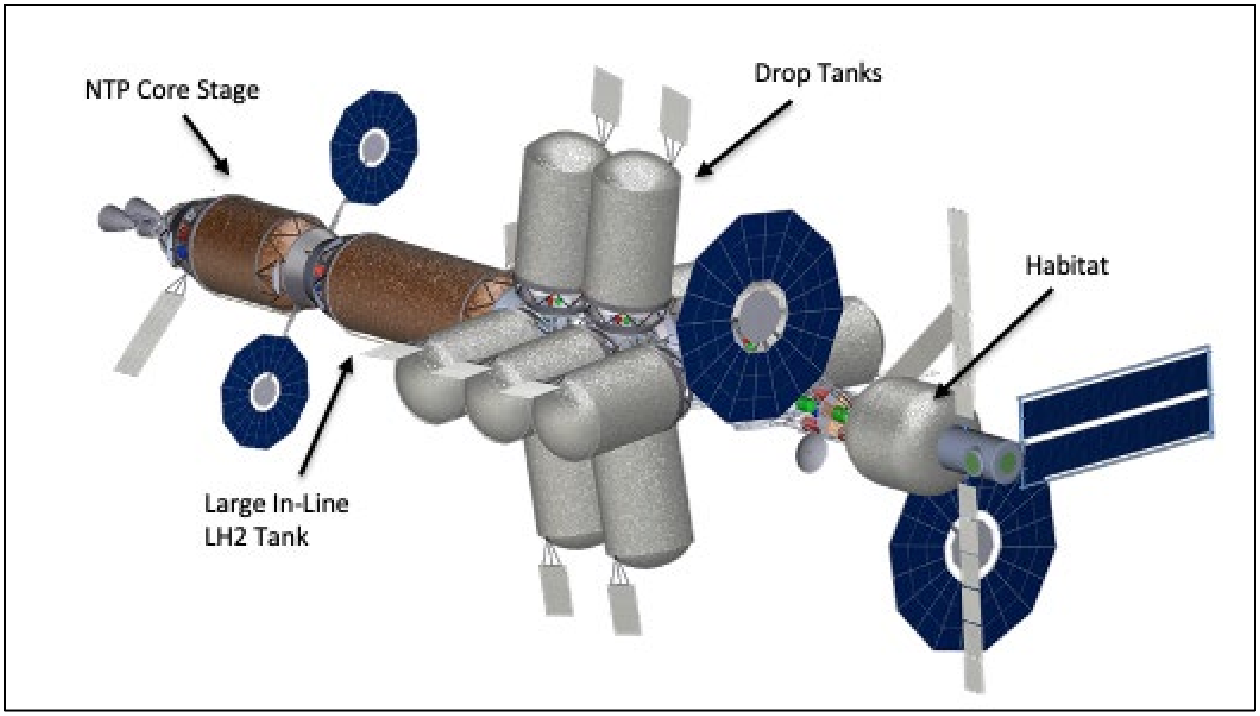

Figure 2. NTP Mars Transportation Vehicle Concept

3. Initial CFM System Assumptions

To maximize consistency in the CFM system design approach between the two vehicle study teams, a set of initial assumptions were developed and provided to each team. These described the CFM technologies to be included in the design, but the teams were given the flexibility to evolve from these initial assumptions to optimize the design. Some of the critical assumptions included:

- Mass of tank internals (e.g., liquid acquisition devices, slosh baffles, thermodynamic vent system (TVS), mass gauge) was estimated as a percentage of metallic tank mass. The percentage used was based on past component designs.
  - With the included liquid acquisition devices, it was assumed that in-space propellant transfer with less than 2% residuals in the supplying tank can be achieved even in zero-g or under moderate adverse acceleration.
  - Propellant gauging is assumed to be accurate to within 2% during “unsettled” (zero-g) conditions and within 1% when the propellant is “settled.”
  - A TVS system was assumed to be required for robust tank pressure control through all mission phases (e.g., launch, aggregation, and operations) if the zero boil-off storage capability was not operating. This requirement may be eliminated as experience with in-space operations is gained.
- The use of spray-on foam insulation (SOFI) depended on the mission concept of operations (e.g., if the tanks were launched dry, no SOFI was required). If they were launched “wet,” an appropriate thickness of SOFI was included.

- For hydrogen, the tank was actively cooled with a two-stage distributed cooling system (i.e., a broad area cooling (BAC) shield), using tubing to carry cooled gas around each tank and appropriate structure and plumbing interfaces to intercept heat before it reaches, or remove heat from, the propellant hydrogen [3].
    - Included a 20 K tube-on-tank stage (helium gas working fluid) and a 90 K tube on shield stage (neon gas working fluid).
    - For LO_2/LCH_4, a single-stage 90 K class BAC on the tank wall was implemented.
    - Note that in Figure 3 the SOFI is shown under the innermost BAC shield. It is also possible to mount the cooling tubes directly to the tank wall with SOFI applied over the top of the tubes. This would be a future design trade that may simplify BAC fabrication and eliminate the thin aluminum shield mass but may also require additional mass of SOFI to ensure adequate thickness over the coolant tubing. It is not anticipated that this change would significantly affect thermal performance during the long duration in-space mission; the SOFI is only serving a purpose while the tank is inside the Earth's atmosphere.
- The insulation for the hydrogen tanks included integrated micrometeoroid and orbital debris (MMOD) protection [2], as shown in Figure 3.

  <u>Inner insulation:</u>
    - A layer of SOFI, if required
    - 10 layers of load-bearing multi-layer insulation (LB-MLI) [4]
    - 90 K BAC shield

  <u>Outer insulation MMOD (47 layers total)</u>
    - 12 MMOD layers
        - 6 layers of Nextel
        - 6 layers of Kevlar
    - 35 reflector layers (LB-MLI)

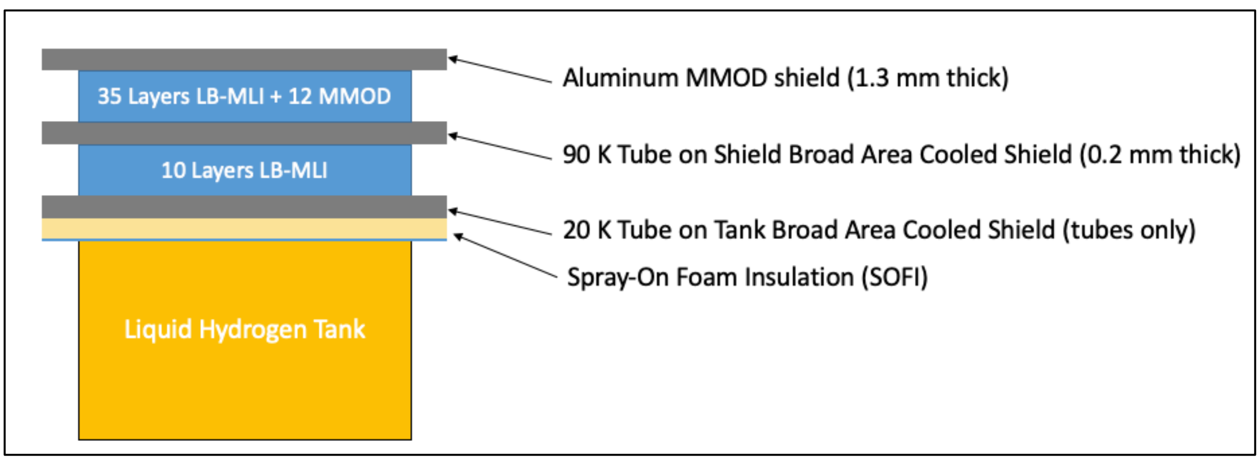

Figure 3. Insulation and Integrated MMOD Protection for LH_2 Tanks

- The insulation for LO_2 and LCH_4 tanks was similar to the approach shown in Figure 3. for the LH_2 tanks, except the 20 K BAC shield and inner 10 layers of LB-MLI were removed.
- For all tanks, an outer aluminum MMOD shield was included [2].

Assumptions regarding the performance of the Reverse Turbo-Brayton (RTB) cryocoolers are summarized in Table 1.

Load Temperature	Lift (cooling capacity)	Specific Power	Specific Mass
20 K	20 W	60 W/W	4.4 Kg/W
90 K	169 W	8 W/W	0.4 Kg/W

Table 1. RTB Cryocooler Performance Assumptions

- Cryogen leakage assumptions
  - Metallic tanks are assumed (eliminates permeability uncertainty) and zero leakage.
  - Ultra-low leakage valve technology:
    - 3-in. valves leak ~0.5 standard cubic inches per minute (SCIM) helium
    - 8-in. valves leak ~1.5 SCIM helium

4. Cryogenic Thermal Systems for LO_2/Methane Stage for NEP/Chem Solution

The Mars NEP/Chem Hybrid spacecraft uses a combination of a nuclear reactor-powered electric propulsion system and a chemical propulsion stage (LO_2/LCH_4 propellant) to transit to Mars from Earth orbit. The thermal system must maintain liquid oxygen and methane tank temperatures below the boiling point to eliminate any propellant boil-off and ensure the propellant can last throughout the mission.

The thermal system accomplishes its task by using a zero boil-off system consisting of a number of cryocoolers, MLI, and the associated system for rejecting waste heat from the cryocoolers. The Compass team performed thermal analysis of the CFM systems using the commercial off-the-shelf (COTS) software Thermal Desktop with RadCAD and SINDA/FLUINT as well as tools developed in-house.

The spacecraft will travel from Earth (1 astronomical unit (AU)) to Venus (0.74 AU) to Mars (1.54 AU). Figure 4 illustrates the mission and shows the operational sink temperatures at each location. The cryogenic thermal control system is sized to operate within this environment. Solar intensity and view angle, as well as the view to warm bodies (e.g., spacecraft solar arrays and radiators) and operation during shadow, are used to determine the worst-case hot and cold conditions. The worst-case hot conditions will occur in sunlight near Venus with all equipment operating, whereas the worst-case cold will be at Mars in shadow. The aspects of cryogenic thermal control and environment, as well as the system components that were addressed or sized in the design and analysis include:

- Radiator panels
- Thermal control cryogenic tanks and propellant lines, cryocoolers, and zero boil-off system for eliminating propellant boil-off
- Heat pipe system for moving waste heat from cryocoolers to radiators
- Heaters for controlling spacecraft component temperature
- Temperature sensors, controllers, switches, data acquisition
- Heat leak through insulation and insulation pass-through

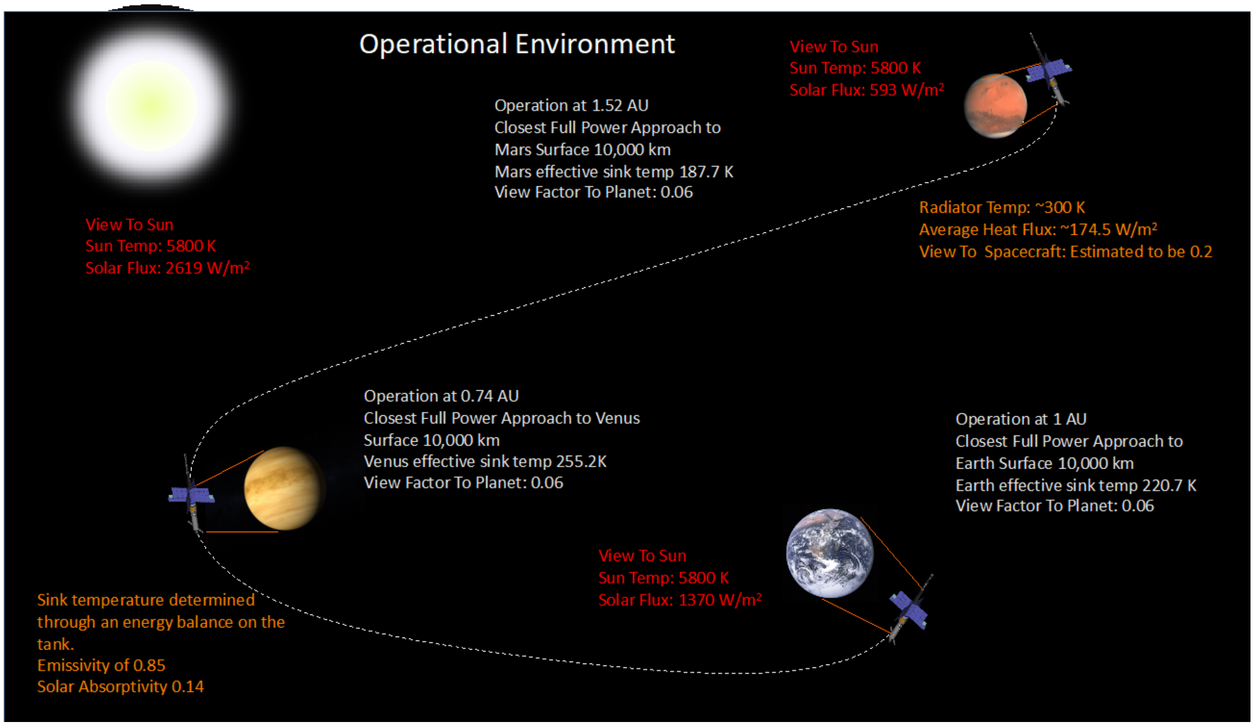

Figure 4. Environmental Thermal Properties Throughout the Mission

Table 2 lists the requirements and assumptions used in sizing thermal system components.

Variable/Component	Value/Description
Nested LO_2/ Liquid Methane Tank	5.0 m Diameter, 14.8 m Total Length (oblate spheroid ends, 11.3 m barrel length)
Waste Heat Load to be Rejected:	Chemical Stage Cryocooler: 338 W at 300 K Chemical Stage Electronics: 2,500 W at 300 K
Operating Temperature	Electronics and Cryocooler: 300 K Rejection Temperature Tank: 90.4 K LO_2 tank (L-Methane tank will be operated at slightly above to avoid freezing the methane)
Multi-layer Insulation (MLI) Used for Electronics	Electronics Section Outer Frame Wrapped in 25 Layers of MLI, Cryogenic Propellant Tank Wrapped in 35 layers of MLI
Environment	Operational Environment: Venus 10,000 km Altitude 0.72 AU (worst-case hot), Mars 1.54 AU (worst-case cold)

Table 2. Vehicle Specification, Requirements, and Assumptions for Thermal System Sizing

The radiator for the cryocooler and electronics within the chemical section use a single-sided body-mounted radiator with variable conductance for moving heat from the electronics and cryocooler hot end to the radiator. The waste heat must be rejected into the surroundings for these components to operate at their desired temperatures. The radiator sizing is based on an energy balance analysis of the area needed to reject the identified heat load into space.

The electronics and cryocooler propellant management thermal control use a body-mounted radiator. This is due to the proximity of the radiator to the heat loads and the available surface area to accommodate the radiator. Using body-mounted radiators near the heat source allows

the use of heat pipes to move the heat from the source to the radiators. The electronics are mounted to the cold plates, where the generated heat is collected. The heat pipes move the heat from the cold plates to the radiator. The number of heat pipe runs depends on the amount of heat to be moved, the capacity of the runs, and the amount of redundancy needed in the system. Heat pipe condenser sections are distributed throughout the back of the radiator, allowing for uniform heating and heat rejection. The radiator is coated to reflect most of the incoming visible solar radiation, significantly reducing the heat load on the radiator.

The thermal control for the cryogenic system is a major aspect of the spacecraft's overall thermal design. In the vacuum of space, radiation heat transfer is the main mechanism for heat leak into the tank from its surroundings. For long-term use in space, the propellant tanks must be more resistant to this heat leak, so MLI was used as the main heat leak barrier. LB-MLI is under development and has been tested to a technology readiness level (TRL) of 5. Its advantages include:

- The support of the BAC shield without the need to pass structural standoffs through the insulation.
- Reduction of the heat flux per layer by up to 56% over traditional MLI [4].

The cryogenic propellants LO_2 and LCH_4 are held in conformal nested tanks. The nested cryogenic propellant tank is broken into two segments, with the LO_2 section held at 90 K and the LCH_4 section maintained at an average ~100 K. The heat leak into the propellant tank using the estimates for LB-MLI from each source is listed in Table 3 for operation near Venus with a sink temperature of 255.8 K. This is the worst-case operating location for the mission and therefore used to determine the heat leak the BAC system must be capable of removing.

Heat Leak Path	90 K Section
Fill Tubes	1.4 W
Structural Supports	15.1 W
MLI (includes passthrough joints, no seam losses)	24.8 W
Total (heat to be removed by cryocoolers)	41.3 W

Table 3. Propellant Tank Heat Leak Summary for Operation Near Venus

Cryocoolers are used to remove the heat loads given in Table 3. The number of cryocoolers required is based on the lift or heat removal capability achievable per cryocooler system. The heat loads were significantly smaller than the cryocooler initial assumptions provided, and a smaller lift capacity was assumed. Along with this change, the specific mass was doubled. Two cryocoolers were included for redundancy. Table 4 lists the cryocooler specifications to meet the total heat removal requirements.

Cryocooler Specifications	90 K
Electrical to Cooling Power Ratio	8 We/Wc
Specific Mass	0.81 kg/Wc
Cooling Lift per Cryocooler	50 W
Cryocooler Mass	40.5 kg
Required Heat Removal (worst case)	41.31 W
Number of Cryocoolers required	2 (1 + 1 Redundant)
Total Cryocooler Power Required	338.8 W

Table 4. Cryocooler Specification

Cryocooler power will vary throughout the mission, with the highest operation near Venus, the location with the worst-case operating conditions.

5. Cryogenic Systems for the NTP Vehicle Hydrogen Tanks

The ACO team also performed thermal analysis of the CFM systems using the COTS software Thermal Desktop with RadCAD and SINDA/FLUINT*,* as well as tools developed in-house. Consistent with initial assumptions, the overall approach to thermal control for hydrogen storage in the ACO study of the NTP vehicle was to leverage two-stage active cooling with 20 K and 90 K cryocoolers, LB-MLI (e.g., 5-mil double-aluminized Mylar with low conductivity spacer stands), and an integral MMOD system. It was assumed the MMOD system would not contribute to thermal performance and each tank would include its own independent cryocooling systems.

Several versions of the Mars NTP vehicle will be required to transport landers and crew and optimize the energy requirements of different opposition-class opportunities. The vehicles will be assembled from various quantities of four propulsion elements, depending on mission requirements. The 7.0 m-diameter Center Core stage and Center Inline LH_2 storage tank were derived from the SLS launch vehicle. The 6.0 m-diameter Side Core stage and Side Inline/Drop LH_2 storage tanks were derived from a commercial launch vehicle. The largest assembled Mars vehicle was the crewed vehicle in the 2039 mission, as shown in Figure 5, with 20 tanks and 90 cryocoolers. Multiple launches are required to place the vehicle elements into Earth orbit for eventual assembly into the Mars transport vehicles. Mission durations will include extended loiters in the aggregation orbit in addition to the two-year Mars mission round-trip time.

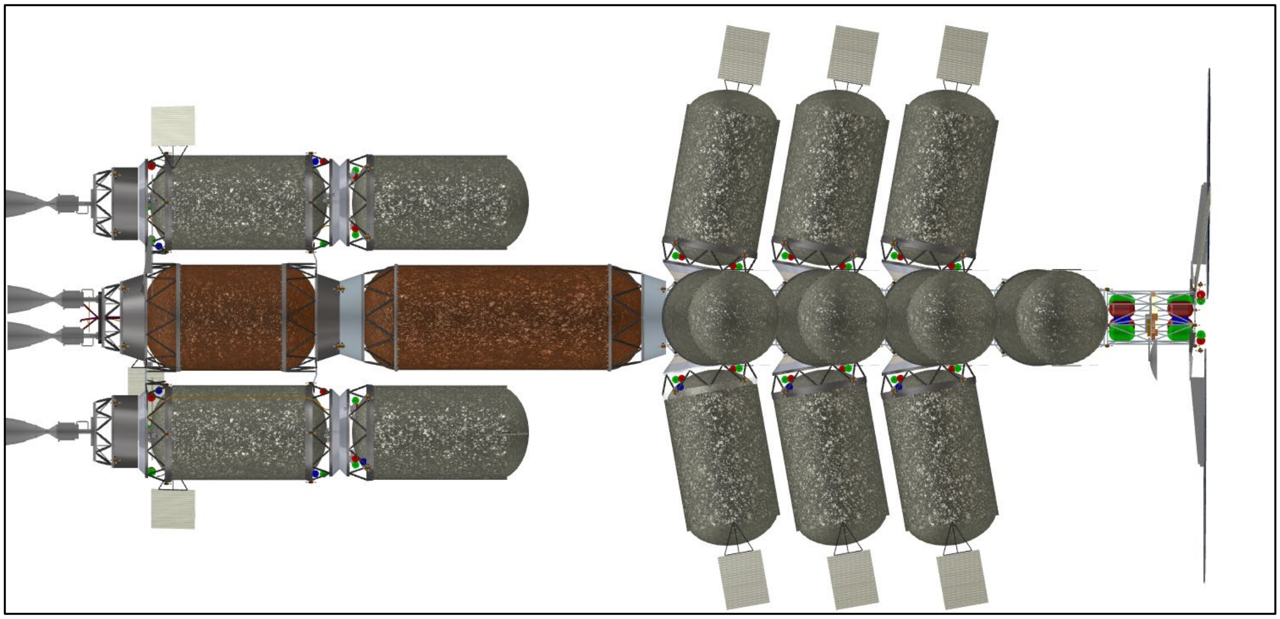

Figure 5. NTP Vehicle Post-aggregation, Prior to Crew Habitat Attachment

All stages and tanks experienced environmental heating. The core stages include nuclear thermal engines, and therefore experienced direct nuclear heating of the bulk propellant as well. Summaries of LH_2 tank environmental and nuclear heating rates are given in Table 5 and Table 6. Note that the combined energy of environmental heating and nuclear heating must be rejected during recovery from the core stages' propellant nuclear heating transient. Although the heating from the nuclear engines indicated in Table 6 is quite high, this heat source is only present for a relatively short period and the amount of hydrogen is large, so the impact is modest. For example, the LH2 in the Center Core Stage tank only increases in temperature by a few kelvin during the "Deep Space Maneuver."

The relevant thermal environments were:

- Perihelion (0.58 AU)
- Earth orbit (2000 x 2000 km)
- Mars orbit (3639.5 x 115889.5 km)
- Mars transit

20 K Active Cooling (W)	Center Core	Center Inline	Side Core	Side Inline/Drop
Insulation System	10.0	16.0	11.4	10.6
Feedlines	2.9	1.3	1.7	1.4
Tank Supports	0.7	0.7	0.6	0.9
20 K Total	13.6	18.0	13.7	12.9
Required Operating 20 K Cryocoolers	1	1	1	1
90 K Active Cooling (W)	**Center Core**	**Center Inline**	**Side Core**	**Side Inline/Drop**
Insulation System	26.9	22.5	22.4	23.5
Feedlines	157.1	25.0	79.6	18.2
Tank Supports	171.3	112.0	90.6	97.0
90 K Total	355.3	159.5	192.6	138.7
Required Operating 90 K Cryocoolers	3	1	2	1

Table 5. ACO Study LH_2 Tank Environmental Heating

Vehicle Element	MPS Maneuver	Heating Rate (kW)	Burn Duration (sec)	Time to Next Maneuver (days)	Required Lift (W)	Required Operating 20 K Cryocoolers
Center Core Stage	Deep Space Maneuver	155.5	4707.4	204.2	42.4	3
Side Core Stage	First Pre-departure Maneuver	51.8	1447.5	20.0	43.4	3

Table 6. ACO Study LH_2 Tank Nuclear Heating

As recommended in the initial assumptions, 20 K and 90 K RTB cryocoolers were used to remove heat from the LH_2 tanks and the bulk propellant or to intercept heat loads before they reached the tanks. The dedicated 20 K and 90 K cryocoolers interfaced with the 20 K and 90 K BAC tubing networks, respectively, at each tank. Dedicated radiators on each tank rejected tank heat loads and cryocooler waste heat to space. The initial assumptions provided for the cryocoolers (section 3) were unchanged for the final design concept. However, input power to 90 K cryocoolers was reduced when analysis determined that less than maximum lift was required. The baseline LH_2 tank insulation system was identical to the initial assumptions, with two clarifications: the spray-on foam insulation (SOFI) layer was 0.75 in. thick, and an outer layer of low absorptivity white paint was applied to the 1.2 mm aluminum MMOD shield. Note that it was not considered practical to use vehicle orientation to reduce heating rates into the LH_2 tanks. Low-conductivity materials were employed for additional thermal control. LH_2 feedlines were made of Inconel 718, 7.0-in. ID leading to the LH_2 tanks and 4.0-in. ID leading to the engines, with 1.0 mm walls. Tank support struts were made of S2 glass with metallic end fittings.

In addition to hydrogen boil-off due to heating, propellant can be lost through leakage, primarily through valve seals. Rough estimates of LH_2 leakage rates over the course of a two-year mission showed no more than 37 kg of LH_2 leakage for a representative number of valves (36 x 3-in. valves, 4 x 8-in. valves), based on gaseous helium (GHe) leakage rates of 0.5 standard cubic inches per minute (SCIM) and 1.5 SCIM for 3-in. and 8-in. valves, respectively. For the ACO

study final design, the vehicle had approximately four times as many valves (many more tanks and interfaces), but not all would be pressurized with hydrogen for the full duration of the mission. For example, once all drop tanks were expended, only the valves on the Core and Inline stages leak hydrogen. Therefore, the total estimate was no more than 148 kg of hydrogen leakage (37 kg x 4) for the ACO study mission. This quantity was determined to be so small relative to the full propellant load of the vehicle that it would have negligible impact on the mission.

6. CFM Technology Development Approach

The conceptual designs of the NEP/Chem Hybrid and NTP Mars transportation vehicles identified the advanced CFM technologies required to successfully store and operate with the cryogenic propellants that enable these challenging missions. A second study objective was to assess the feasibility of maturing the needed technologies in time to support a mid-2030s Mars mission. Toward this goal, the CFM individual technologies' maturity and integrated system maturity were assessed, and a notional approach was developed to address critical gaps. Figure 6. illustrates the scope of this plan. For this assessment, the targeted "Technology and Subsystem Maturation" performance goals aligned with the technology assumptions outlined above in section 3. Initial CFM Assumptions. In addition to addressing individual technology and subsystem development, critical gaps exist in development and validation of high-fidelity predictive analytical tools and successfully proving the technologies and integrated CFM systems in the microgravity and thermal environment of space. A key aspect of developing the needed predictive capabilities for design and performance analysis will be developing appropriate datasets of microgravity effects on fundamental fluid and thermal processes through careful, detailed diagnostics, and microgravity experiments. NASA is funding major elements of this approach, including CFM flight demonstrations through Tipping Point contracts, ground and parabolic flight experiments, advanced model development, and the Zero-Boil-Off Tank (ZBOT) series of ISS experiments. Further, additional confidence gained from conducting more capable missions with cryogenic propulsion is anticipated through the Human Landing System Program. The technology development approach developed in this study demonstrates that with continued investment, the necessary advanced CFM technologies can be ready to support crewed Mars missions in the mid-2030s.

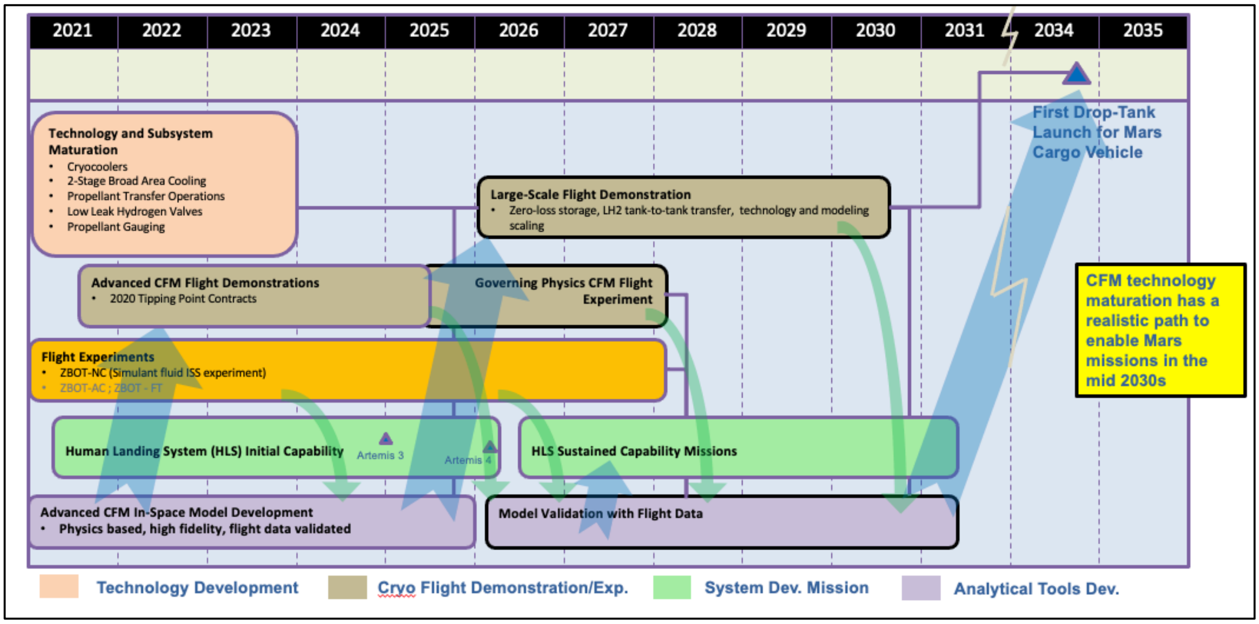

Figure 6. Notional Advanced CFM Technology Development Approach

7. Summary

NASA's vision for crewed exploration includes returning to the moon with much more capability to study the moon and demonstrate the technologies needed for missions to Mars. The vision then expands to taking crew to Mars in the 2030s. Through this study of NEP/Chem Hybrid and NTP propulsion options for opposition-class crewed missions to Mars, the required advancements of CFM technologies were identified. Further, it was shown that with continued investment, development of the needed CFM technologies and CFM system capabilities is feasible in a timeframe to support Mars missions in the mid-2030s.

Acknowledgements

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